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MEMORANDUM REPORT

for the

Army Air Forces, Materiel Command

ANALYSIS OF THE HIGH-ALTITUDE COOLING OF THE

RANGER SGV-770 D-4 ENGINE IN

THE BELL XP-77 AIRPLANE

By Jack N. Nielsen and Lloyd E. Schumacher

Langley Memorial Aeronautical Laboratory
Langley Field, Va.

October 20, 1943

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SUMMARY

As a part of a general NACA investigation of the cooling of the Bell XP-77 airplane, an analysis has been made of the cooling installation in the high-altitude XP-77 airplane for the entire altitude range. The analysis has been based on heat-transfer data taken in the propeller-research tunnel on the Ranger SGV-770 C-1B engine. This engine was tested with the same cylinders, aluminum fins, and turbulent-flow baffles proposed for the high-altitude SGV-770 D-4 engine. The cowling pressure measurements used in the analysis were obtained in the propeller-research tunnel on the low-altitude version of the XP-77 airplane equipped with an NACA designed cowling. Other data have been supplied by the Bell Aircraft Corporation and Ranger Aircraft Engines. The most recent methods developed by the NACA at LMAL have been used in making the altitude predictions.

The calculations show that the pressure drop available is not sufficient to cool the engine in climb at military power above 15,000 feet. The low available pressure is due to the low airplane speed and the high air-flow losses.

Cooling in cruising power at 27,000 feet will be marginal. The high required pressure drops are due to the low barrel-flange temperature limit.

The smallest conventional oil cooler with 9-inch tubes 0.256 inch in internal diameter that will keep the oil inlet temperature to the engine below 185° F will have a diameter of 11 inches.

The modified lower lip of the NACA cowling has too low a critical speed for satisfactory operation in high-speed level flight at altitude. Further modifications have been made to the test airplane to remedy this defect.

INTRODUCTION

Late in 1942 the NACA at the request of the Army Air Forces and in close cooperation with the Bell Aircraft Corporation and Ranger Aircraft Engines began a general wind-tunnel investigation of the cooling of the Bell XP-77 airplane. A full-scale model of the airplane equipped with the low-altitude Ranger SGV-770 C-1 engine was tested in the propeller-research tunnel. The effects of several cowling configurations on the available cooling pressures and the air-flow losses were studied. In addition a heat-transfer correlation, based on the method of reference 1, was made for the C-1B engine with aluminum fins and turbulent-flow baffles. The final results of the cooling pressure measurements are presented in reference 2, and the final results of the heat-transfer correlations are presented in reference 3.

The general layout of the cooling installation in the XP-77 airplane has been described in reference 2. The power plant is an inverted-vee, 12-cylinder, air-cooled engine. Air for carburetion, engine cooling, and oil cooling enters the cowling directly behind the propeller. This arrangement permits full utilization of the slipstream pressure in cooling the engine. The low-altitude XP-77 airplane will be equipped with a Ranger SGV-770 C-1B engine. The C-1B engine has the aluminum fins and turbulent-flow baffles described in reference 2. The high-altitude XP-77 airplane will be equipped with a Ranger SGV-770 D-4 engine, which has a Planiol type supercharger. The D-4 engine will have the same cylinder and baffles as the C-1B engine.

The XP-77 airplane was tested in the propeller-research tunnel with both a Bell designed cowling and an NACA designed cowling. On the NACA designed cowling the cooling-air inlet was enlarged and reshaped. The engine air outlet was moved farther forward and equipped with guide vanes. The engine-cooling analysis and the critical-speed analysis are made from pressure measure-

ments on this improved cowling.   After the tests in the
propeller-research tunnel, the airplane was moved to the
full-scale tunnel for further tests.   Here the inlet of
the cowling was modified to raise its critical speed, the
oil-cooler duct was relocated, and the oil-cooler flaps
were removed.   The oil-cooling analysis is made for this
arrangement.

The purpose of this report is to predict the per-
formance of the cooling installation in the high-altitude
airplane from the measurements made in the propeller-
research tunnel and from manufacturers' data.   Calcu-
lations have been made for all operating altitudes to
determine the pressure drops required for engine cooling
and oil cooling, the pressure drops available for cooling,
the outlet areas for the oil cooler and engine, and the
critical speed of the cowling.   The heat-transfer data
obtained for the SGV-770 C-1B engine in the propeller-
research tunnel were extrapolated to altitude by the
latest method developed by the NACA at LMAL.   Altitude
data on the estimated airplane performance and the
estimated engine-operating conditions were furnished by
the Bell and Ranger companies, respectively.   The
calculations for the oil cooler were based on experimental
heat-transfer and pressure-drop data for circular oil
coolers with tubes 9 inches in length and 0.256 inch in
internal diameter.   Army summer air temperatures and
densities were used for all the calculations.

COOLING ANALYSIS

The Bell corporation has furnished calculated curves
of the airplane speed against altitude for the different
power conditions.   These curves, which are used to
determine the available pressures, are shown in figure 1.
The curve for cruising at 70 percent normal power has
been estimated from the other curves.

The engine brake horsepowers and engine speeds for
all operating altitudes are plotted in the engine
specifications of the Ranger company.   The values are
summarized in table I.   Ranger Aircraft Engines have
also furnished the engine charge-air flows and fuel-air
ratios given in table II, and the intake manifold pres-
sures and temperatures given in table III.   All engine
data have been marked preliminary and are not for final
design purposes.

Table III shows that the intake manifold pressure and temperature remain about constant between sea level and 12,000 feet for military power. Accordingly it has been assumed that there is no change in intake manifold conditions from sea level to 12,000 feet for the other powers. The adiabatic efficiency of the supercharger calculated from the data in table III is about 70 percent for the high-blower conditions. This value has been used in calculating the intake manifold temperatures for all other high-blower conditions. A further assumption is made that the intake manifold pressure and engine charge-air flows remain constant with altitude for constant power.

Engine-cooling calculations.- The engine-cooling calculations have been made with the help of the heat-transfer data presented in reference 3 for the SGV-770 C-1B engine with the aluminum fins and turbulent-flow baffles. The available pressures and air-flow losses have been taken from reference 2 for the NACA designed cowling, designated in the reference as configuration II. These data, taken at sea level, are sufficient for predicting the cooling at altitude.

A sample calculation of the pressure drop, $\sigma\Delta p$, required at sea level to maintain the average head temperature at a specified limit is given in reference 3 to illustrate the use of the heat-transfer correlations. This value of $\sigma\Delta p$ may also be used to determine the mass flow of cooling air from the sea-level engine air-flow calibration. This procedure may also be used to determine the engine-cooling-air mass flow at altitude. However, at altitude the value of $\sigma\Delta p$ itself must be corrected to account for the pressure-drop increases due to the decrease in density of the cooling air.

A simplified explanation showing why increased pressure drop results from the decrease in density of the cooling air during its passage through the engine is as follows: From the law of continuity a decrease in density causes an increase in velocity. An increase in pressure drop is necessary to produce this acceleration of the cooling air. The higher velocity causes greater skin-friction losses and hence an additional increase in pressure drop. Under sea-level conditions the density changes are usually negligible but under high-altitude flight conditions the changes are appreciable and should not be neglected. As previously mentioned, these corrections were made in the computations.

Engine specifications require that the spark-plug gasket or barrel flange must not exceed certain maximum allowable temperatures. The following simple method for determining whether the spark plugs or barrel flanges require a greater engine pressure drop was used for these calculations. The determination is made with the help of the curve of maximum spark-plug temperature against average spark-plug temperature and the curve of maximum barrel-flange temperature against average spark-plug temperature plotted in figure 2. These data have been taken from reference 3. The values of the average spark-plug temperatures corresponding to the maximum allowable spark-plug and barrel-flange temperatures in table IV are read from figure 2. The engine pressure drop to cool the engine to the lower average plug temperature is then obtained with the help of the heat-transfer correlation. The cooling pressure drops required to keep the hottest spark-plug gasket or the hottest barrel flange below the temperature limit have been computed for a range of altitudes from sea level to 27,000 feet and for cruise, normal, military, and take-off powers.

From the temperature limits in table IV and from figure 2 it was found that at sea level the flanges determine the pressure drop required for take-off and cruise powers, and that the spark plugs determine the pressure drop required for normal power. For military power the flanges require slightly more pressure drop than the spark plugs. The experimental curves of figure 2 are strictly applicable only at sea level. Calculations show that the relationship between the maximum spark-plug temperature and the average spark-plug temperature should not vary greatly with altitude so that the experimental sea-level curve may be used for all altitudes without much error. The change in the relationship between the maximum flange temperature and the average spark-plug temperature with altitude, however, cannot be neglected if the cooling calculations are made for very high altitudes. For take-off power, which goes up to only 9000 feet, this change was neglected, but for cruising power the change was taken into account by using the heat-transfer correlation for the flanges.

The available pressure drops across the engine were determined from the curves of inlet total pressure against thrust coefficient and inlet pressure losses against inlet-velocity ratio given in reference 2. A minimum static pressure behind the engine of about $-0.1q_0$

was obtained with flaps.   Changes in the angle of attack caused minor variations in this pressure, and these variations were neglected in determining the pressure drop available.   No decrease in airplane speed due to flap drag was taken into consideration.   The total pressure increases in the cowling inlet due to the propeller slipstream are higher than would ordinarily be expected from a propeller with a normal load distribution.   This increase in propeller loading is probably due to the low axial velocity ahead of the inlet.   These gains were measured using a low-altitude propeller and may not be fully realized with the high-altitude propeller.

The results of the engine pressure calculations are given in figure 3(a) for level flight and in figure 3(b) for climb.   The curves show that at 27,000 feet and at cruising power there is a margin of about two inches of water between the pressure drop available and the pressure drop required.   However, figure 4(a) shows that the flaps are nearly full open for this condition so that flap drag will probably reduce the airplane speed enough to eliminate this margin entirely.   The high values of the required pressure drop in cruise are due to the low barrel-flange temperature limit.   Sufficient pressure drop is available in all other level-flight conditions to cool the engine without opening the flaps.

The curves of figure 3(b) show that in climb at military power sufficient pressure drop will not be available above 15,000 feet.   Cooling in climb at high altitude is inherently difficult because of the low available pressures.   In this case the low available pressures at altitude are due to high air-flow losses as well as to a decrease in the total pressure of the slipstream.   The abrupt rise in the required pressure drop at the first critical altitude and its rapid increase thereafter are characteristic of supercharged engines with no inter- cooling.

Fin and baffle changes may be very helpful in alle- viating the cooling problem in climb.   For instance, in climb at 15,000 feet, the air-flow losses up to a point just within the turbulent-flow baffles amount to about $0.6q_o$.   The data of reference 2 show that about three- quarters of this loss occurs when the air turns sharply into the baffles.   Redesigning the baffles to reduce this loss may allow the airplane to cool to a higher altitude.   Further improvements may be gained by modi- fying the cowling to reduce the inlet and outlet losses.

Although the engine will have marginal cooling at
15,000 feet with the flaps full open, it may be possible
to cool at a higher altitude by sacrificing rate of
climb for climbing speed.

Calculations have been made to determine the outlet
areas for the engine air in both level flight and climb.
The outlet areas are based on the measured pressures in
the outlet of the NACA flaps of about free-stream static
pressure for the flap-flush position and $-0.4q_o$ for
the flap-full-open position. The total pressure losses
behind the engine and the decreased density of the heated
air in the outlet were also taken into consideration.

The calculated values of the total engine-air outlet
areas are presented for the level-flight conditions in
figure 4(a). The variation in the outlet static pressure
with flap position was neglected for take-off, military,
and normal powers because the flaps are close to the flush
position. The flaps will be nearly full open in cruise
at 27,000 feet so that in this case the variation in
outlet static pressure with flap position may not be
neglected. A linear variation in the outlet static
pressure from the flap-flush to the flap-full-open posi-
tion was assumed in obtaining the outlet area for cruise
above 15,000 feet.

The calculated values of the outlet area for climb
are presented in figure 4(b). Again a linear variation
in the outlet static pressure and outlet losses with flap
position between the flap-flush and the flap-full-open
positions was assumed. From the figure it may be seen
that the engine will cool in climb at military power up
to about 15,000 feet where the flaps will be full open.
Little altitude would be gained by using a larger outlet
area.

Oil-cooling calculations.- Oil-cooler calculations
have been made to determine the required pressure drops,
the available pressure drops, and the required outlet
areas. The heat-rejection characteristics for the oil
coolers are given in figure 5 and the pressure-drop
characteristics are given in figure 6. These figures
are taken from manufacturers' data on circular oil
coolers with round tubes 9 inches in length and
0.256 inch in internal diameter. The oil coolers in
figure 5 were obtained with air at sea-level temperature.

The use of air at very low temperature (about -20° F) would have resulted in lower values of the heat-transfer rates in figure 5 because of oil congealing. However, oil congealing will not be serious in Army air at 27,000 feet altitude, although on a cold winter day it may have an appreciable effect at the same altitude. The pressure drops of figure 6 refer to the static pressure drop across the oil cooler.

In the calculations the maximum heat rejections have been based on the figure of 1700 Btu per minute given in the engine specifications for normal-rated power. For other brake horsepowers the maximum heat rejections have been taken proportional to the brake horsepower as shown in table V. An oil-flow rate of 100 pounds per minute has been used for all the computations because the amount of cooling air for a given heat rejection and oil-inlet temperature to the engine does not vary appreciably with the oil-flow rate in the normal operating range of the oil cooler. A maximum inlet-oil temperature to the engine of 185° F has been given in the engine specifications.

The required pressure drops for oil cooling have been determined with the help of figures 5 and 6, taking into account the temperature rise of the cooling air due to adiabatic compression. A small correction has been made to the pressure drop to account for the same density effects mentioned in connection with the engine pressure drops.

The values of the available pressure drop for oil cooling have been computed in the same manner as those for engine cooling. The total pressure in the inlet has been obtained from the propeller thrust coefficient and a small inlet loss has been subtracted from the inlet total pressure to get the total pressure at the oil-cooler face. The inlet losses, which are only a few percent of the free-stream dynamic pressure, were measured on two 6-inch oil coolers located beneath the engine cam boxes at cylinders 3 and 4. Finally, to get the maximum possible total pressure drop across the oil cooler, a total pressure of $0.2q_o$ behind the oil cooler was assumed since a sliding door rather than a hinged flap is under consideration for the oil cooler.

The calculations show that the oil-cooling require-
ments will be a maximum at sea level in level flight at
take-off power.   In this condition the oil cooler must
dissipate 2440 Btu per minute per 100° F temperature
difference between the average oil and entering air.
Interpolation between the curves for the 9- and 11-inch
oil coolers in figure 5 reveals that a 10-inch oil
cooler will not be quite large enough to handle this
oil-cooling load.   Accordingly, complete calculations
have been made only for the 11-inch oil cooler.   For
comparison, however, some calculations have been
included for a 9-inch oil cooler.

The calculated values of the required and available
pressure drops for level flight at various powers are
shown in figure 7(a) for the 11-inch oil cooler and in
figure 7(b) for the 9-inch oil cooler.   The 11-inch oil
cooler will have sufficient cooling capacity at all
operating altitudes and power conditions, but the 9-inch
oil cooler will not have sufficient cooling capacity for
normal or military powers below 10,000 feet.   Under these
conditions the 9-inch oil cooler is operating under heavy
load and great increases in the air-flow rate will not
greatly increase the heat rejection.   In climbing flight
the situation is the same as in level flight.   As
figures 8(a) and 8(b) show, the 11-inch oil cooler has
sufficient pressure available for cooling at all alti-
tudes and powers, but the 9-inch oil cooler does not have
sufficient pressure available below 7000 feet for military
power or at any altitude for take-off power.

The rapid rise in the required oil-cooler pressure
drop near sea level, particularly for high-speed level
flight, is due to the adiabatic temperature rise of the
cooling air.   Near sea level the temperature difference
between the cooling air and the oil is least, and the
effect of adiabatic temperature rise is greatest.   This
effect is illustrated by figure 9(a), where the ratio of
the oil-cooler air mass flow to the engine-cooling air
mass flow is plotted for level flight and various powers,
and by figure 9(b) for climbing flight.   In high-speed
level flight near sea level the ratio of oil-cooler air
mass flow to engine-cooling air mass flow is nearly 1.
The general trend of the ratio is to decrease with alti-
tude.

For pete's sake.

sure! the available ΔT at sea level is less with oil cooler than cylinders.

The outlet areas for the 11-inch oil cooler have been computed assuming that the static pressure in the sliding-door outlet is free-stream static pressure. The calculated values are given in figure 10 for both level and climbing flights. A maximum area of 0.47 square foot is required at sea level for climb at take-off power, while a minimum area of about 0.065 square foot is required for cruise at high altitudes. At 27,000 feet the required outlet areas have not yet started to increase with altitude.

Inlet-velocity ratio and critical-speed determination.- The cowling inlet-velocity ratios have been calculated from the flow rates of the engine charge air, the engine cooling air, and the 11-inch oil-cooler air. An inlet area of 1.21 square feet has been used since this is the area of the revised inlet of the XP-77 airplane being currently tested in the full-scale tunnel.

The inlet-velocity ratio tends to decrease a few percent between sea level and 12,000 feet as shown in figure 11. After the change in blower speed at about 12,000 feet, the inlet-velocity ratio increases gradually with altitude. For level flight the inlet-velocity ratio lies between the limits of 0.31 and 0.44, and for climb it lies between the limits of 0.64 and 0.76.

For this airplane in level flight the inlet-velocity ratios for military and normal powers at 27,000 feet are 0.41 and 0.39, respectively. From pressure measurements on the modified lower lip of the NACA designed cowling for an inlet-velocity ratio of 0.39 and for a lift coefficient of 0.15 (reference 2) the critical Mach number is found to be about 0.59 using the von Kármán-Tsien method of extrapolation. At 27,000 feet in standard air a critical Mach number of 0.59 corresponds to an airspeed behind the propeller of 405 miles per hour. Since the level-flight true airspeeds given for military and normal powers at 27,000 feet by figure 1 are 422 and 407 miles per hour, respectively, the modified lower lip of the NACA cowling will be operating above its critical speed for these conditions. On the revised cowling being tested in the full-scale tunnel the lower lip has been refaired to raise its critical speed.

CONCLUDING REMARKS

1.   There is not sufficient pressure drop available above 15,000 feet for cooling the engine in climb at military power.   Since large pressure losses occur when the air turns sharply into the turbulent-flow baffles, it may be possible to cool at a higher altitude if the baffles can be redesigned to reduce these losses.   Some smaller gains may be realized by further cowling modifications to reduce the inlet and outlet losses.

2.   The cooling for cruise power at 27,000 feet will probably be marginal.   In this condition there is a difference of only 2 inches of water between the available and required pressure drops and the flaps will be partially extended.   The flap drag will result in a reduced airspeed or will require an increased power to maintain the same airspeed.   In the first case the cooling will be more difficult because of the lower available pressure drop and in the second case the cooling will be made more difficult because of the greater required pressure drop.   The primary reason for the unusually high required pressure drop is the low barrel-flange temperature limit.

3.   An 11-inch circular oil cooler with tubes 9 inches in length and 0.256 inch in internal diameter will have sufficient capacity to meet the heat-rejection specifications of the Ranger SGV-770 D-4 engine.   The inlet oil temperature to the engine may be kept below 185° F without the use of oil-cooler flaps if the heat rejections do not exceed those in table V.

4.   At 27,000 feet the modified lower lip of the NACA cowling may be operating above its critical speed of 405 miles per hour since the predicted level-flight airspeeds for military and normal powers are 422 and

407 miles per hour, respectively.   The critical speed
of the lower lip may probably be raised appreciably by
modifying its contour.

Langley Memorial Aeronautical Laboratory,
        National Advisory Committee for Aeronautics,
            Langley Field, Va., October 20, 1943.

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Approved:

        John W. Crowley, Jr.,
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    Cooled Engine Cylinders.   Rep. No. 612, NACA, 1938.

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3. Conway, Robert N., Emmons, M. Arnold, Jr., and
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    Oct. 22, 1943.

TABLE I

ENGINE PERFORMANCE - RANGER SGV-770 D-4

Condition	Brake horsepower	Altitude range (ft)	Rpm
Take-off	575	Sea level to 9,000	3400
Military	515	Sea level to 12,000	3400
Military	500	12,000 to 27,000	3400
Rated	465	Sea level to 12,000	3200
Rated	450	12,000 to 27,000	3200
Cruise	325	Sea level to 27,000	2800

TABLE II

SEA-LEVEL ENGINE AIR FLOWS AND FUEL-AIR RATIOS - RANGER SGV-770 D-4

Brake horsepower	Fuel-air ratio	Charge-air flow (lb/hr)	Rpm
575	0.108	4450	3400
515	.107	3900	3400
427	.086	2950	3400
365	.080	2600	2600
306	.081	1800	2200

TABLE III

PLANIOL SUPERCHARGER PERFORMANCE ON

RANGER SGV-770 D-4 ENGINE

Standard air

[40° F has been subtracted from the manifold mixture temperature to allow for gasoline evaporation.]

Power condition	Manifold mixture temperature, (°F)	Manifold pressure (in. Hg)	Altitude (ft)
Take-off	115	41	Sea level
Military	108	36	Sea level
Military	108.5	35.6	12,000
Military	198.5	41.4	22,500
Rated	93	31.8	12,000
Rated	162	35.0	23,000
Cruise	102	26.0	25,000

TABLE IV

MAXIMUM ALLOWABLE SPARK-PLUG GASKET TEMPERATURES

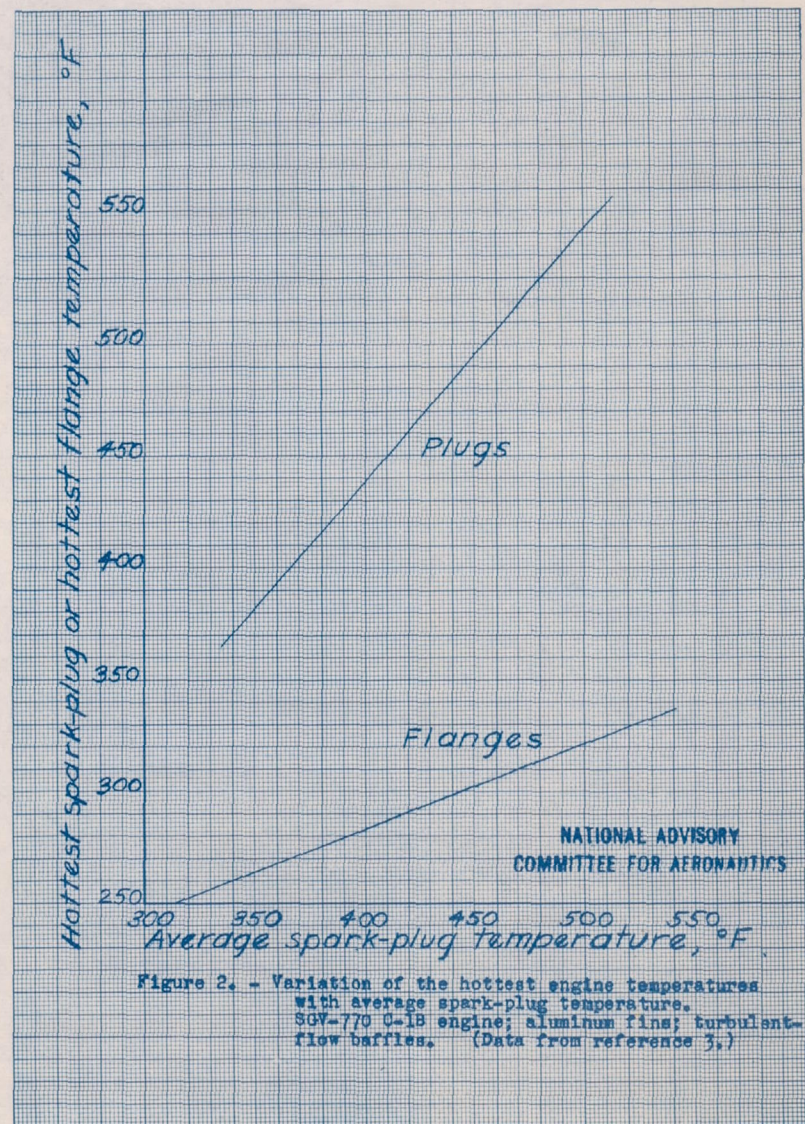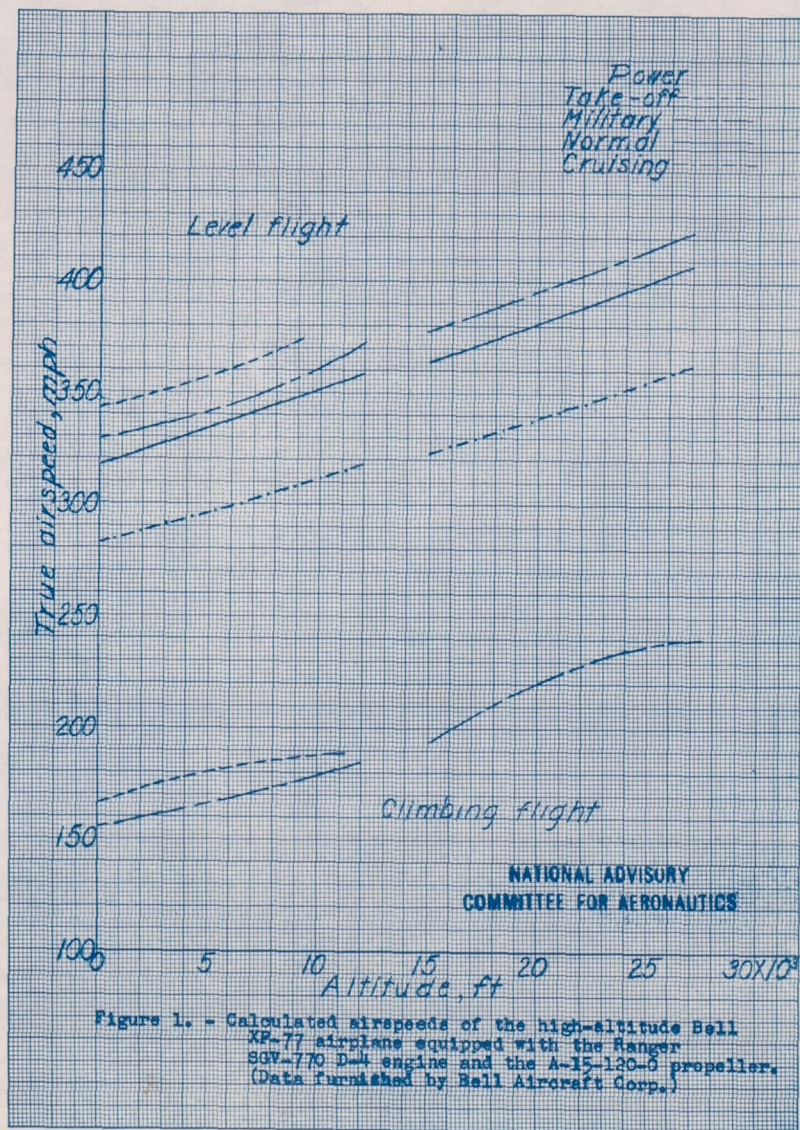| Condition	Spark-plug-gasket temperature limit		Barrel flange temperature limit		Time
	(°F)	(°C)	(°F)	(°C)	
Take-off power	518	270	302	150	5 minutes
Normal-rated power	482	250	302	150	Continuous operation
70 percent normal-rated power	464	240	266	130	Continuous operation
Military	482	250	302	150	5 minutes

TABLE V

ASSUMED OIL-COOLING REQUIREMENTS

RANGER SGV-770 D-4 ENGINE

[Oil-flow rate, 100 lb/min]

Brake horsepower	Heat rejection (Btu/min)	Oil temperature rise (deg)
465	1700	34
575	2100	42
515	1880	38
325	1190	24

Figure 1. - Calculated airspeeds of the high-altitude Bell
XP-77 airplane equipped with the Ranger
SGV-770 D-4 engine and the A-15-130-0 propeller.
(Data furnished by Bell Aircraft Corp.)

Figure 2. - Variation of the hottest engine temperatures
with average spark-plug temperature.
SGV-770 0-1B engine; aluminum fins; turbulent-
flow baffles.   (Data from reference 3.)

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Power
Take-off ------
Military --------
Normal --------
Cruising --------

Pressure drop, in water

60

50

40

30

Δp available

20

10

0

Δp required

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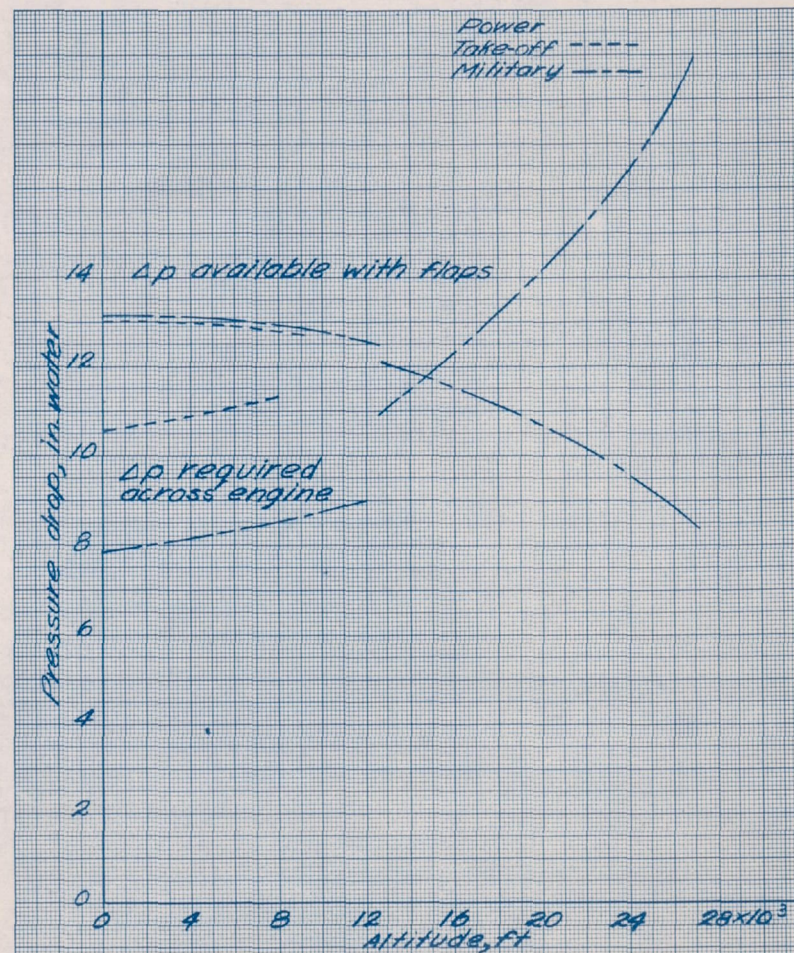COMMITTEE FOR AERONAUTICS

0   5   10   15   20   25×10³
Altitude, ft

(a) Level flight.

Figure 3. - Calculated required pressure drop and available
pressure drop for cooling SGV-770 D-4 engine.

Power
Take-off ------
Military --------

Pressure drop, in water

14   Δp available with flaps

12

10   Δp required
     across engine

8

6

4

2

0
0   4   8   12   16   20   24   28×10³
Altitude, ft

(b) Climbing flight.

Figure 3. - Concluded.

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Power
Take-off ----
Military ——
Normal ——
Cruising —·—

Flaps full open

Flaps flush

Outlet area, sq ft

Altitude, ft

(a) Level flight.

Figure 4.- Calculated engine-air outlet areas in NACA cowling.

Power
Take-off ----
Military ——

Flaps full open

Flaps flush

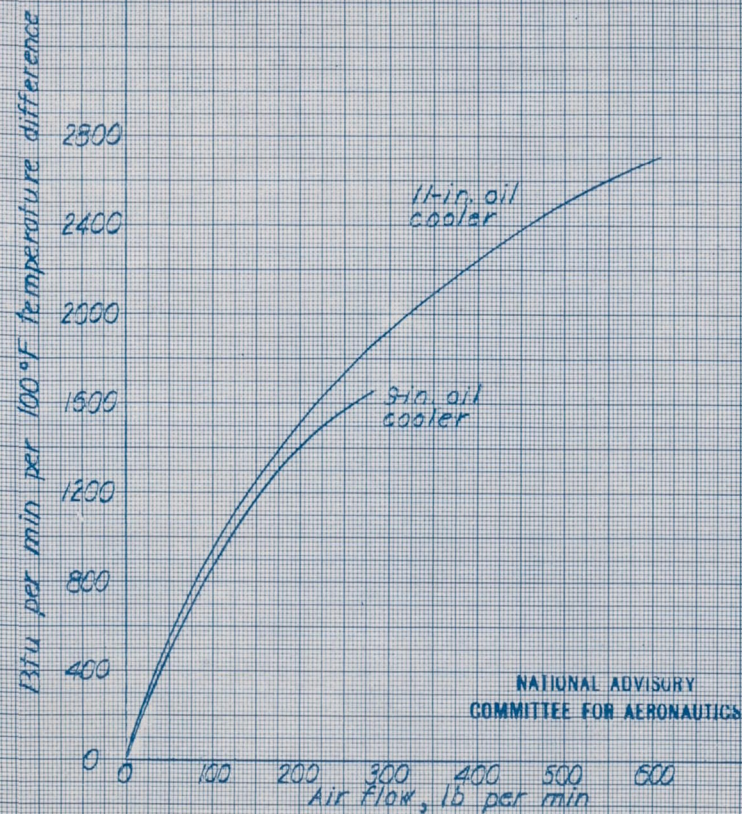Outlet area, sq ft

Altitude, ft

(b) Climbing flight.

Figure 4. - Concluded.
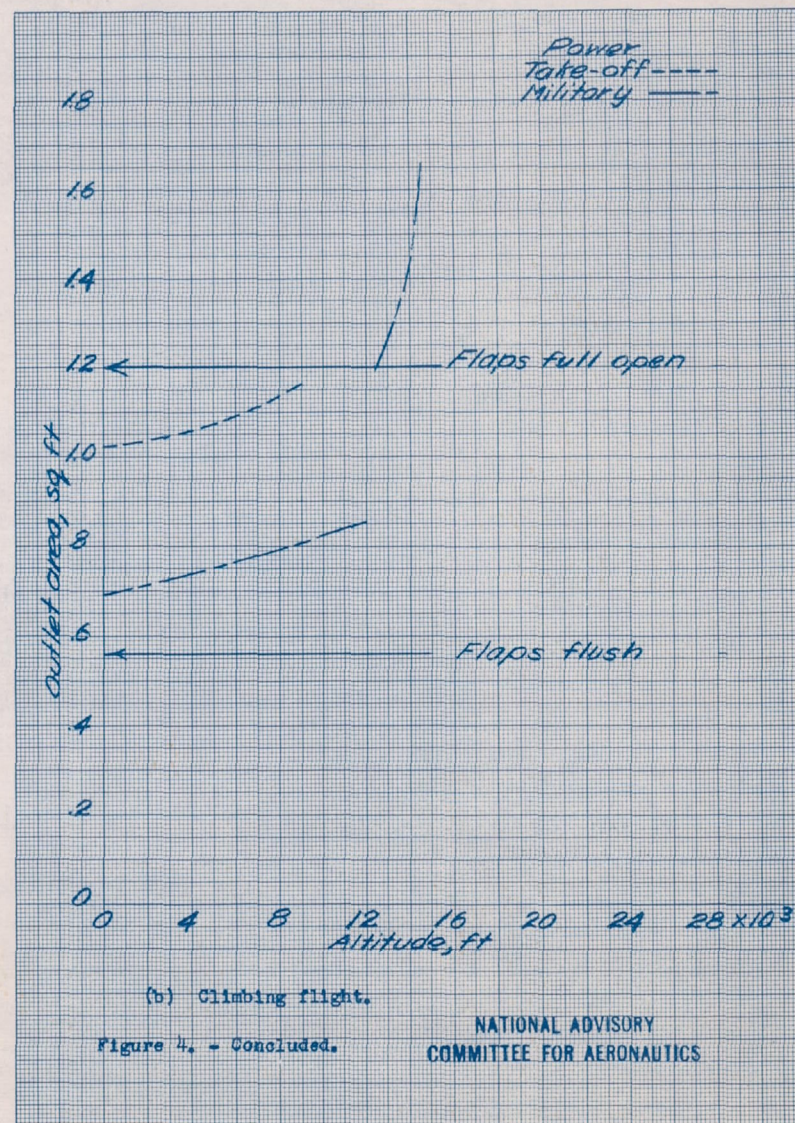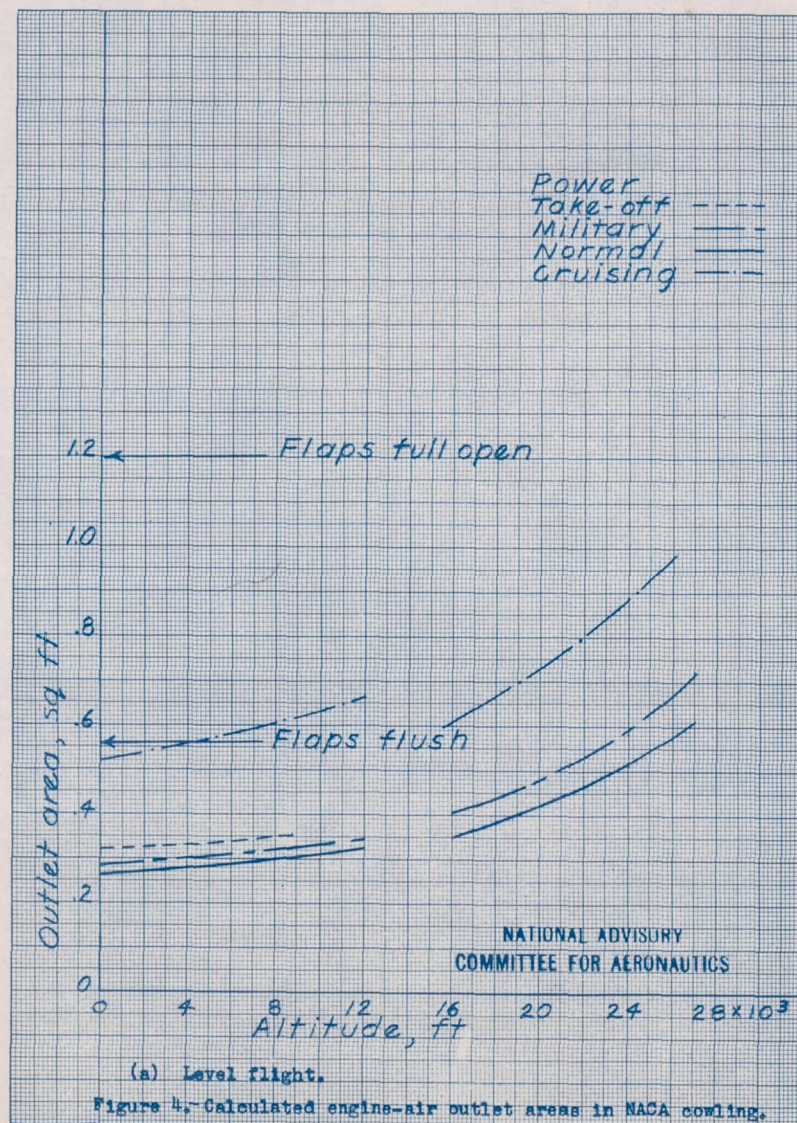
Figure 5. - Heat rejection for 100° F difference between average oil and entering air temperatures for round oil coolers from manufacturers data; tube length, 9 inches; internal diameter, 0.256 inch; oil-flow rate, 100 pounds per minute.
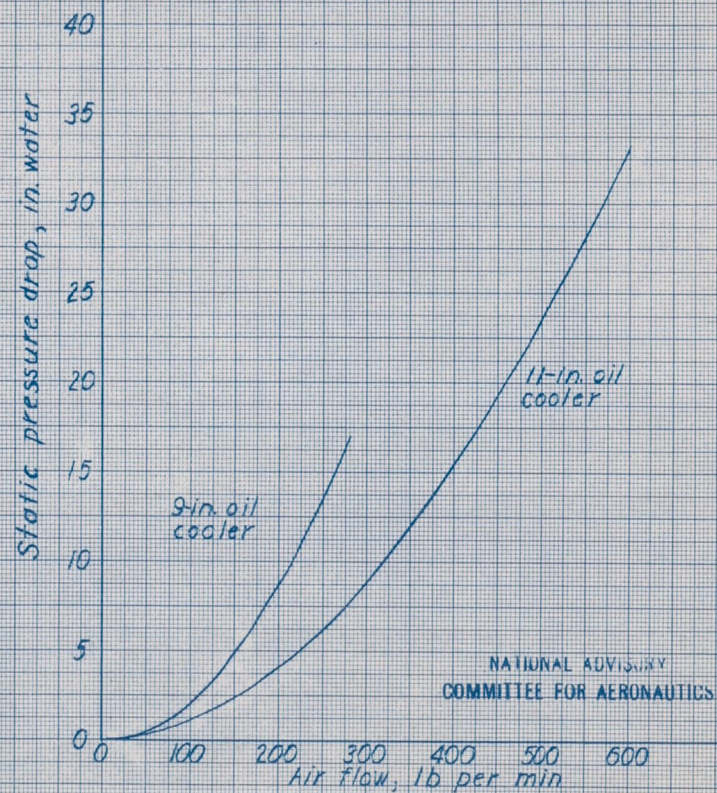
Figure 6. - Static pressure drop characteristics of round oil coolers from manufacturers data; tube length, 9 inches; internal diameter, 0.256 inch.
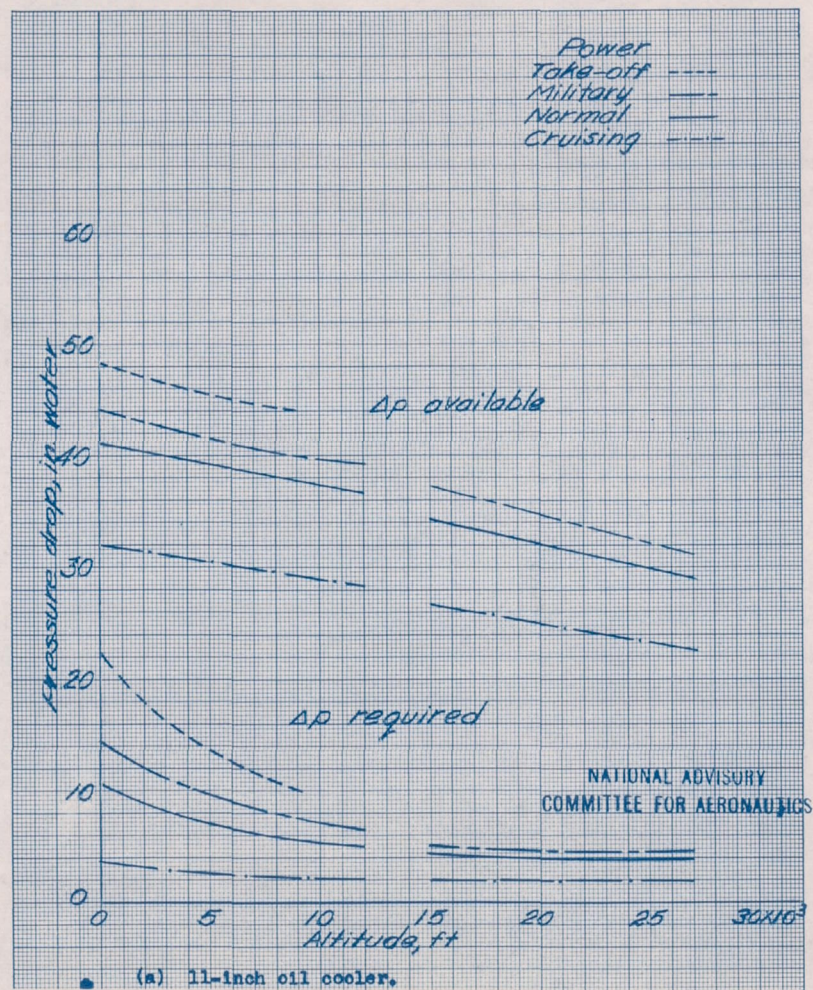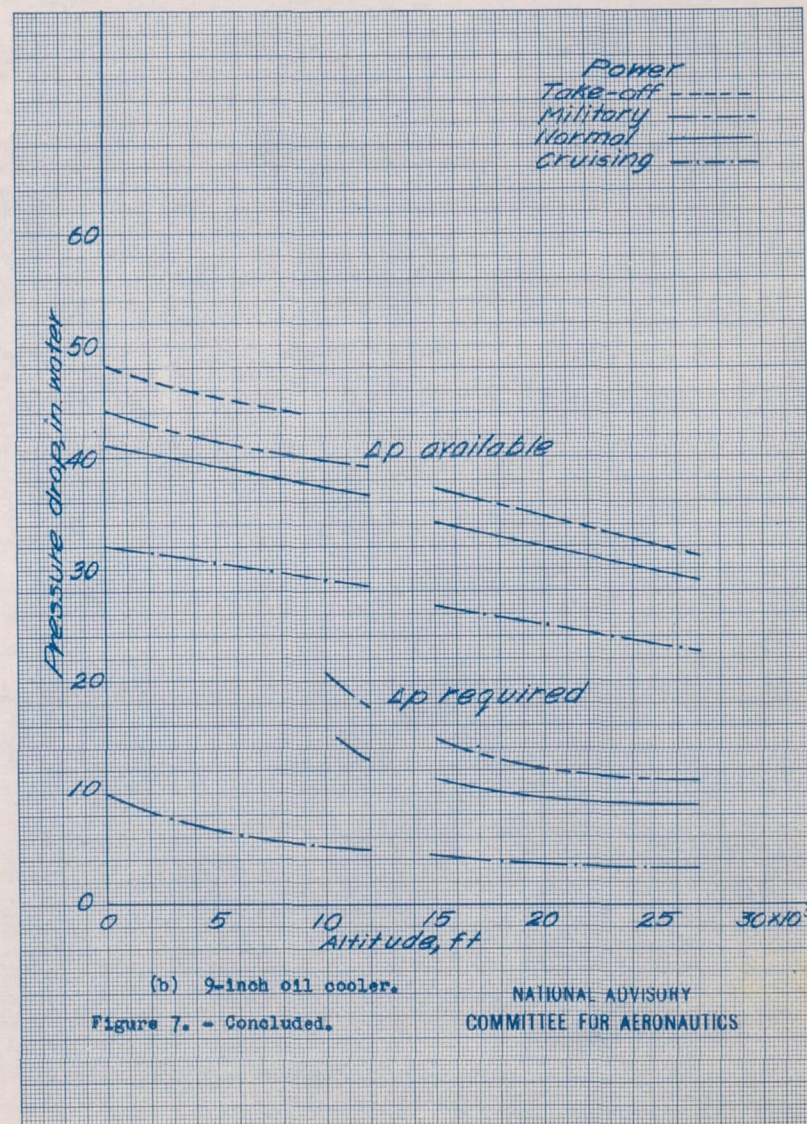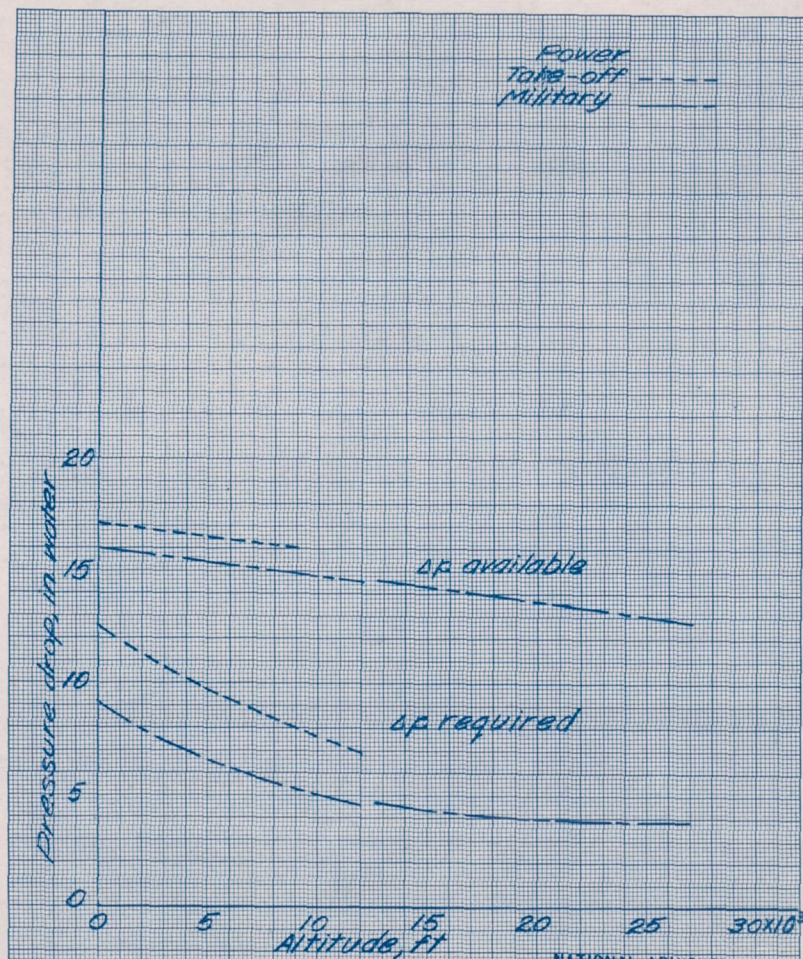
(a) 11-inch oil cooler.

Figure 7. - Calculated required pressure drop and available pressure drop for oil cooling of SGV-770 D-4 engine in level flight; oil inlet temperature, 185° F.

(b) 9-inch oil cooler.
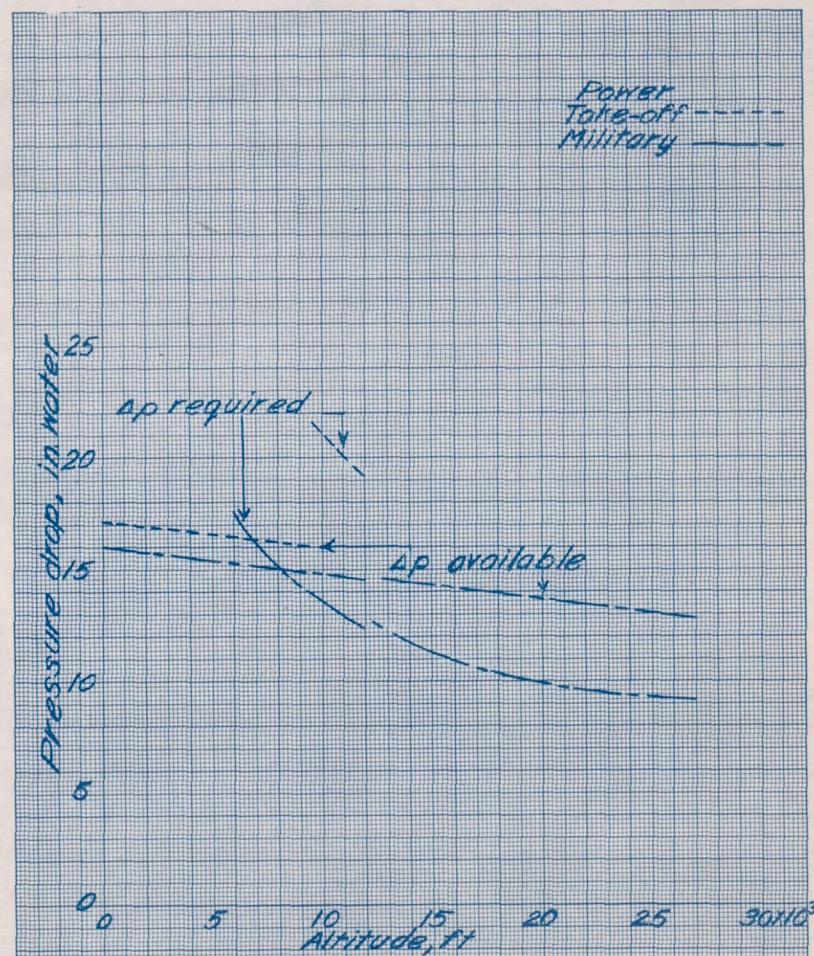
Figure 7. - Concluded.

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(a) 11-inch oil cooler.

Figure 8. – Calculated required pressure drop and available pressure drop for oil cooling of SGV-770 D-4 engine in climbing flight; oil inlet temperature, 185° F.
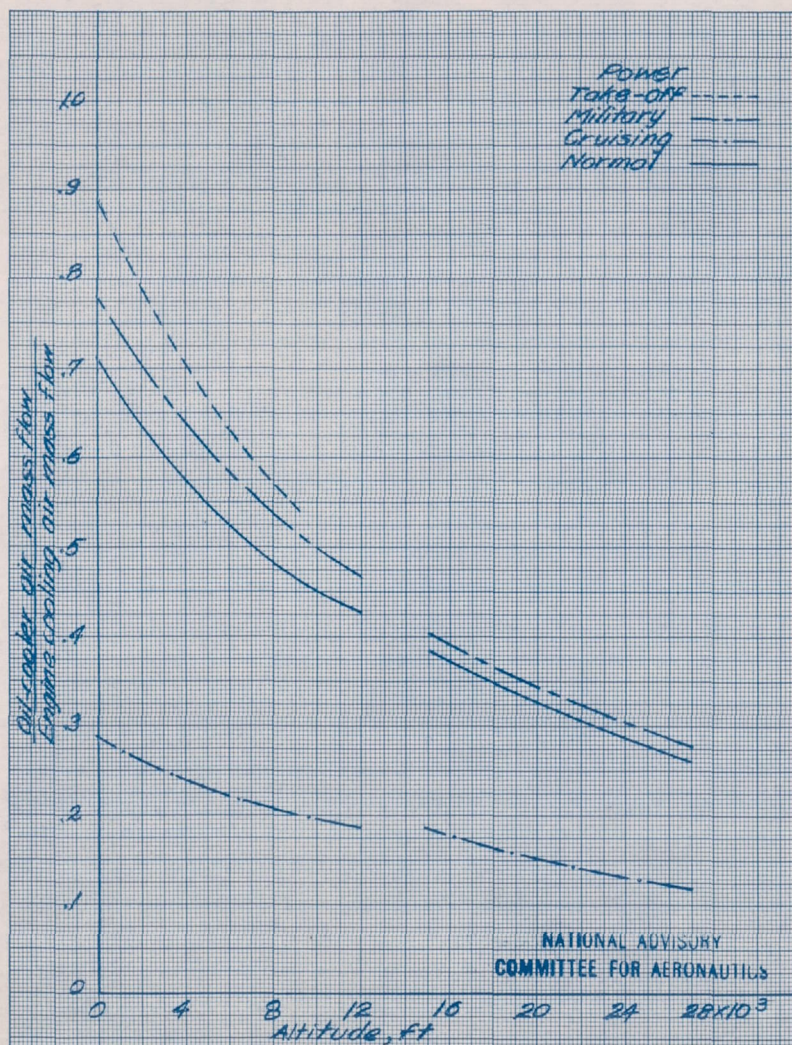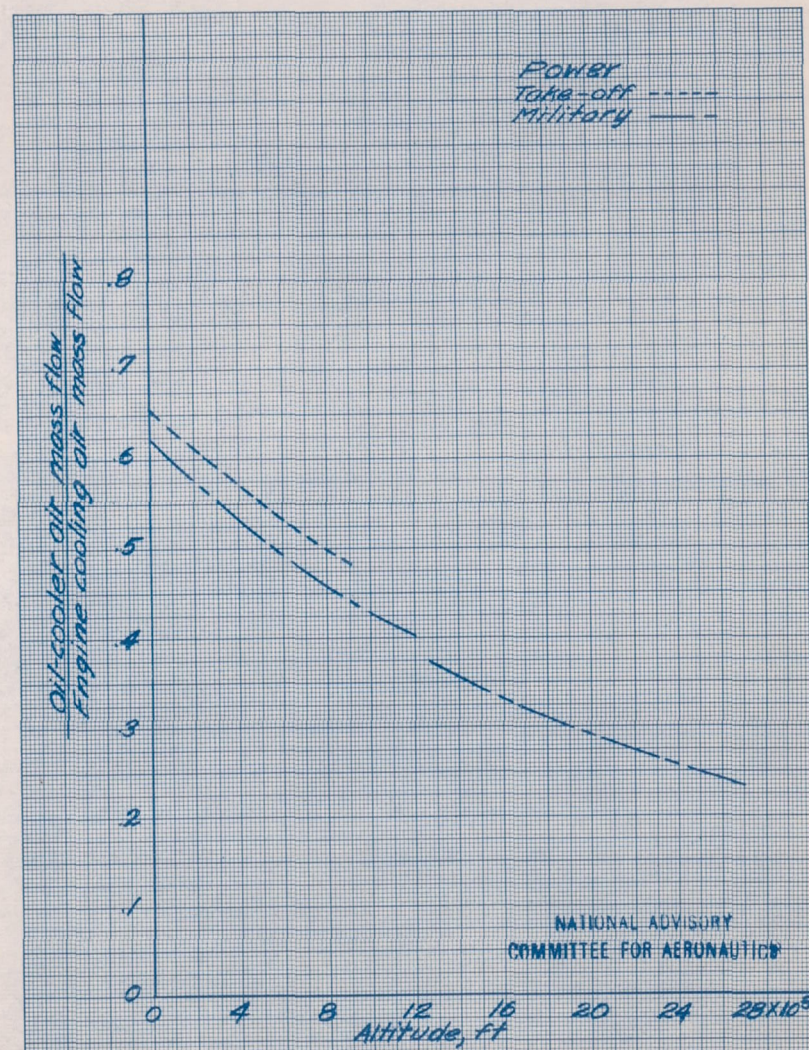
(b) 9-inch oil cooler.

Figure 8. – Concluded.

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Power
Take-off -----
Military —— ——
Cruising —— · ——
Normal ————

Oil-cooler air mass flow / Engine cooling air mass flow

Altitude, ft

(a) Level flight.
Figure 9. - Calculated ratio of 11-inch oil-cooler air mass
flow to engine-cooling air mass flow.

Power
Take-off -----
Military —— ——

Oil-cooler air mass flow / Engine cooling air mass flow

Altitude, ft

(b) Climbing flight.
Figure 9. - Concluded.

Figure 10. — Calculated oil-cooler air outlet area for 11-inch cooler; oil-inlet temperature, 185° F; no flaps.

Figure 11. — Calculated cowling-inlet-velocity ratios; inlet area, 1.21 square feet; 11-inch oil cooler.